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DESIGN AND TEST OF A BORON -ALUMINUM HIGH TEMPERATURE WING

R. J. Richey, Jr. and T. E. Hess Aircraft and Crew Systems Technology Directorate NAVAL AIR DEVELOPMENT CENTER Warminster, Pennsylvania 18974

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The feasibility of utilizing	the high bucklir	ng stability characteristics
of boron - aluminum advanced composite material in a simple, low-cost spar-		
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carry the primary bending and torsion loads, mechan		
light gage steel sub-structure, wh		

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SUMMARY

The feasibility of utilizing the high buckling stability characteristics of boron-aluminum material in a simple, low-cost spar-rib-skin construction for a thin airfoil structure has been investigated for high temperature application up to 589 degrees K. A weight saving of 30% in comparison to the production article is projected in this boron-aluminum version of the BQM-34E wing, while increasing its temperature capability to 589 degrees K.

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INTRODUCTION

The emphasis in current Naval aircraft structural development is on reduction of weight and cost and improvement of performance. In addition, as flight speeds increase and lift augmentation and thrust vectoring are utilized in Vertical-Short Takeoff/Landing (V/STOL) aircraft, high-temperature structures may be required to withstand the effects of aerodynamic heating and hot exhaust gases. Significant achievements have been made in reducing structural weight by utilizing composite materials, i.e., boron or graphite-epoxy, for moderate-temperature applications, up to 450 degrees K. Similar improvements for higher service temperatures, up to 589 degrees K, require the use of graphite/polymide or boron-aluminum materials.

Despite its current high cost, which is expected to be significantly reduced as usage increases, boron-aluminum has many advantages. It has higher longitudinal stiffness and strength than steel and greater room-temperature transverse and shear stiffness than titanium, while its density is less than that of aluminum. In addition, it has high bearing strength and retains the high thermal and electrical conductivity and weldability of its aluminum matrix.

The objectives of this program were to develop a high-temperature (589 degrees K) composite structural design applicable to thin lifting surfaces, and to demonstrate the concept in a primary aircraft structural component.

Normal design practice for a thin aerodynamic surface, which is being considered here, would be to use full depth honeycomb sandwich construction. However, for high temperature applications, bonding of the skins to the honeycomb core becomes a problem. It was the intent of this program, therefore, to investigate the feasibility of stabilizing the skins with discrete stiffeners at a reasonable cost and weight.

The design which was developed in this program consists of variable thickness boron-aluminum skins, to carry the primary bending and torsion loads, mechanically fastened to a light stainless steel substructure, which resists transverse shear and stabilizes the skins. The viability of the concept depends on whether this stabilization of the skin can be accomplished with a practical number and spacing of substructure elements. Fabrication cost and complexity were minimized by using simple shapes and conventional metal forming and fastening methods. The demonstration article chosen is the wing of the BQM-34E remote-piloted vehicle whose maximum thickness is only three percent of its chord, Figure 1.

Information from material and structural tests has been utilized in the evolution of the wing design. Experimentally verified material stiffness and strength properties have been incorporated into the analysis, together with buckling criteria which have been modified as a result of subcomponent development tests.

DESIGN REQUIREMENTS AND CONSIDERATIONS

The design of the B/Al version of the BQM-34E wing is based on production wing static strength, stability and flutter requirements. The critical flight load condition dictating the design, results from a 5g symmetric pull-up at R.T. An additional design requirement, a 4g symmetric pull up at 589°K, was specified for the B/Al prototype wing.

The high temperature requirement necessitated the selection of thermally compatable materials to be used in the wing design. Specifically, the coefficient of thermal expansion for the light gage metal supporting substructure had to closely match that of the B/Al skins to minimize thermal stresses at elevated temperatures. Stainless steel (TH1050) which is structurally adequate at 589°K and thermally compatible with the B/Al laminate skins was selected as a satisfactory material for the substructure. Both materials have a thermal expansion coefficient of approximately $11.0\,\mu$ m/m °C.

Stiffness requirements dictate that the wing exhibit flutter free behavior in the flight regime ranging from Mach 1.1 at sea level to Mach 3.0 at 23600 m (60000 ft.).

B/A1 WING - FINAL DESIGN OVERVIEW

WING CONFIGURATION

The profile of the B/Al version of the BQM-34E wing duplicates that of the production metal wing. A low cost design approach was followed by approximating the actual wing aerodynamic contour with a simplified wedge shape. Referring to Figure 2, all chordwise wing sections are constant depth closed out with simple wedge leading and trailing edge pieces. Spanwise, the wing tapers linearly from root to tip. Across the center wing box the skins are allowed to assume their natural pure bending curvatures.

SKINS

The basic skin configuration for the B/Al wing design, shown in Figure 3, consists of B/Al tension and compression skin pieces with tailored $(0^{\circ}, \pm 45^{\circ}, 90^{\circ})$ ply construction. Because of the B/Al laminate fabrication diffusion bonding process which involves a multi-step pressing operation, the B/Al main wing skins were kept to a manageable size by incorporating a wing center line skin splice. The joining is accomplished with a single stainless steel splice plate (2.54 mm, (.1 in.)) and a double row of mechanical blind fasteners (4.76 mm (3/16 in.)). Also, separate B/Al trailing edge pieces and stainless steel sheet leading edge pieces are spliced to the main skins along substructure spars.

Both main skins are step tapered, with gradual ply build up toward the wing centerline, optimized to satisfy critical flight load requirements. The final laminate design for the skins was arrived at through iterative stress analysis and experimental specimen and subcomponent testing. The final ply scheme for the tension and compression B/Al skins is schematically

shown in Figures 4 and 5. Skin laminate design drawings are attached at the end of the report. Both tension and compression skins are four plies (.108 cm) at the wing tip, with ply build up to 13 plies (.352 cm) and 16 plies (.168 cm) respectively, across the overall wing box. The extra plies are added to the compression skin to satisfy buckling requirements. Also, both skins are locally built up to 24 plies (.640 cm), in the area of high stress adjacent to the aft attachment of the wing.

SUBSTRUCTURE

Considering only half the wing, referring to Figure 6, the main elements of the light gage stainless steel substructure include seven spars, a tip and root rib and five wing/fuselage bolt attachment fittings. The spar and rib elements are mainly channels, with gages varying from .052 cm (.020 in.) to .127 cm (.050 in.) depending on design requirements. The spar elements run along constant percent of chord lines and are tapered linearly from wing root to tip. The five wing/fuselage attachment fittings tie the substructure elements together along the wing/fuselage bolt attachment lines. Forward spars extend from the fittings across the wing box. The flanges of the wing box spars are separate angle pieces rolled to match the curvature of the skins. The angles are internally spot welded to web sheets to form channel elements.

FASTENING

Fastening of all the structural elements is accomplished with rivets. Standard stainless steel .476 cm (3/16 in.) dia. solid rivets in conjunction with shear clips are used to fasten the substructure elements together. Fastening of the B/Al skins to the substructure is accomplished with .476 cm (3/16 in.) dia. stainless steel blind fasteners. Double rows of blind fasteners in conjunction with .476 cm (3/16 in.) stainless steel plates, as shown in Figure 7, are used to splice the upper and lower half skins together at the wing center line. Similar splice designs are used to connect leading and trailing edge pieces to the main wing skins.

ANALYSIS

NASTRAN

Stress analysis of the B/Al wing design was accomplished by constructing a finite element model, and running a series of NASTRAN static analyses, for the critical 5g maneuver load condition, optimizing the design. The tension and compression wing skins were modeled with quadrilateral and triangular plate elements which have both inplane and bending stiffness. B/Al laminate constitutive relationships used in the NASTRAN analysis were determined from basic laminate theory using the material property constants of unidirectional B/Al. The substructure spars and ribs were modeled with bar elements with shear properties built in. Because of wing symmetry only half of the wing needed to be modeled. The model configuration including grid point and element identification is shown in Figures 8 through 10.

Bulk data for the NASTRAN model is included in Appendix A. Maximum tension and compression skin limit load stresses obtained from NASTRAN for the final laminate design are shown in Figures 11 and 12 respectively.

BUCKLING ANALYSIS

The boron aluminum wing compression skin was sized to satisfy buckling requirements by using NASTRAN stresses in conjunction with standard orthotropic simply supported plate theory. Since the skins are mechanically fastened to the substructure the simply supported boundary condition is a conservative assumption. Buckling loads were calculated for the most highly stressed compression skin NASTRAN elements in each discrete skin gage region. Several iterative cycles were needed to size the skin for buckling stability. Table 1 lists the final results for the compression skin buckling analysis. The critical buckling load due to compression, Nxcr, and the critical buckling load due to shear loading, Nxycr, are compared with the loading the laminate must withstand at design ultimate, Nxult and Nxyult. Margins of safety in buckling due to combined compression and shear loading were calculated using the relation

$$M.S. = \frac{2}{R_L + \sqrt{R_L^2 + 4R_S^2}} - 1$$

where:

$$R_{L} = \frac{N_{xult}}{N_{xcr}}$$

$$R_S = \frac{N_{xyult}}{N_{xycr}}$$

Although the margins of safety for ultimate load were slightly negative for several of the compression skin elements, they were considered acceptable at this point since the analysis was conservative and testing was planned to assess the accuracy of the analysis method. Also, when considering design limit loading, all margins of safety would be positive.

DYNAMIC ANALYSIS

A NASTRAN real eigenvalue run was made to obtain normal mode data for the B/Al wing design. Based on the results of this run and the fact that the B/Al wing design is both stiffer and has less mass than the production wing, the wing was assumed to be flutter free and a rigorous flutter analysis of the B/Al wing was not included in the design cycle.

EXPERIMENTAL TESTING

INTRODUCTION

In order to experimentally validate design procedures and establish a design criteria on which to base the final B/Al full scale wing design, a

series of coupon specimens and two major subcomponents were fabricated and tested. The testing phase of the program included only room temperature testing. This was justified because the critical flight load condition is the R.T. 5g maneuver. To save on fabrication cost 4130 steel was substituted for the stainless, in all subcomponent substructural members.

COUPON SPECIMENS

A number of B/Al coupon specimens including tension and rail shear were tested to validate the material properties used in the design of the full-scale wing. The specimen configurations are shown in Figure 13. A summary of the coupon test results run at NADC are shown in tables 2 through 4. Results of tensile specimen tests run by Americom, Inc. on the basic B/Al laminates used in the tension and compression wing skin design are shown in Table 5. Results of these tests were satisfactory, ultimate loads and material properties in some cases were slightly lower than available standard B/Al properties.

BOX BEAM SUBCOMPONENT

Design

In order to evaluate the manufacturing processes intended for construction of the full-scale wing and to verify the buckling capability of the B/Al compression skin, a box beam specimen representative of the aft wing box region as shown in Figure 14 was designed, fabricated and tested. The aft wing box region was selected for experimental investigation because the compression skin is buckling critical in this area and a box beam type specimen presents minimum fabrication complications and can be symmetrically loaded to facilitate testing.

The box beam specimen, shown in Figure 15, which has a span of 107 cm (42 in.) and a width of 18 cm (7 in.) incorporates the same basic design features as found in the actual aft wing box. The detailed engineering drawing of the box beam is included in the foldouts. The box beam center span between the attachment bolt hole center lines, like the actual wing, is 45.7 cm. (18 in.). The center span substructure channels are constructed of 7.62 mm (.030 in.) rolled 4130 steel angles, to form constant radius flanges, spot welded to a 12.70 mm (.050 in.) 4130 web sheet. The box beam extension arm substructure channels are brake formed and follow a constant spanwise taper. The box beam incorporates eight load fittings, four representative of the aft wing/fuselage attachment fittings and four outer corner load fittings for testing. The compression skin is .267 cm, 10 ply boron/ aluminum with $0^{\circ} \pm 45^{\circ}$, $0^{\circ} \pm 45^{\circ}$, 0° ply orientation. To reduce cost the tension skin is .254 cm (.1 in.) gage stainless steel since only the buckling capability of the B/Al compression skin is of interest. All box beam structural elements and skins are assembled with mechanical fasteners.

Instrumentation

The boron/aluminum wing box beam specimen was instrumented with axial strain gages and strain rosettes as diagrammed in Figure 16. The gages were positioned to monitor spanwise bending and shear stress distribution in both tension and compression skins, stress concentration around the bolt holes and initiation of buckling in the compression skin.

Loading

The box beam was loaded at the eight load fitting bolt holes to produce a condition of pure bending in the center section. This condition with total ultimate applied load of 38.6 kN approximates the critical 5g maneuver load condition. The box beam test set up is shown in Figure 17.

Test

After several initial load cycles to 30% D.L.L. to exercise the specimen a run to failure was made. Buckling of the B/Al compression skin initiated at a load of 288.0 kN comparing well with analysis based on simply supported orthotropic plate theory which predicted initiation of buckling at a load of 314.1 kN. The early onset of buckling may be attributed to actual B/Al compression skin material properties being somewhat lower than those used in the analysis. The specimen continued to sustain increased loading after onset of buckling up to 612.9 kN, at which catastropic failure occurred. The failure is shown in Figure 18. The results of this test were used to substantiate the full-scale compression wing skin design for buckling stability.

WING SUBCOMPONENT

Design

In order to evaluate the behavior of the wing design in the area of highest tensile and compressive stresses, which is adjacent to the aft wing-to-fuselage attachment location, a second development test specimen was designed, fabricated, and tested. This was a subcomponent, outlined in Figure 19, which contained significant design details of the actual wing, with some minor alterations to simplify its fabrication and to provide test load application.

The tension and compression B/Al skins maintain constant ply thickness of 13 and 16 plies respectively over the entire subcomponent surface area. The ply orientation scheme of the skins is identical to that of the full-scale wing's center section. The boron-aluminum skins were fabricated by Amercom, Inc., including the countersunk holes which were made by electric discharge machining, Figure 20.

The substructure parts shown in Figure 21 which stabilize the skins at a constant depth of 4.10 cm were made and assembly operations performed at NAVAIRDEVCEN. At the subcomponent root end the wing center section skin splices are accurately represented by double row rivet attachment to .476 cm (3/16 in.) steel splice plates. These splice plates are supported

by a solid aluminum spacer bar which allows the complete assembly to be clamped for a cantilever test load set up. At the subcomponent free end, a 2.54 cm (1.0 in.) Al plate is fixed for test load application. The complete subcomponent assembly is pictured in Figure 22. The detail design drawings for the subcomponent are attached in the foldouts.

Test Loading and Instrumentation

Test loads to be applied to the B/Al wing subcomponent were determined with the aid of a NASTRAN loads analysis. This analysis resulted in a set of test loads which when applied to the subcomponent produced a stress field in the B/Al skins similar to the stress field present in the actual full scale wing skins when subjected to the 5g maneuver load condition.

The test set up shown in Figure 23 consists of the subcomponent mounted to a strongback testing facility; loads were applied to the specimen through two independent sets of wiffle trees by manually operated hydraulic jacks.

The subcomponent was instrumented with 73 strain gages and three deflection transducers. The gages monitor critically stressed regions on both tension and compression skins and are also paired internally and externally on the compression skin to check for initiation of buckling as shown in Figures 24 through 26.

The test load procedure was as follows:

- 1. Apply 30% D.L.L., 10% increments, check strain and deflection data.
- 2. Apply 50% D.L.L., 10% increments, check strain and deflection data, re-apply 50% D.L.L., 2 cycles.
- 3. Apply 100% D.L.L., 10% increments, check strain and deflection data, re-apply 100% D.L.L., 4 dycles.

Test Results

After initial loading to 30% D.L.L. strain and deflection data was plotted. Referring to Figures 27 and 28, typical strain and deflection vs. load plots from the test data reveal nonlinear, inelastic behavior exhibited by the B/Al skins. The second applied load cycle to 50% D.L.L. yielded approximately linear elastic response in the skins up to the previously applied load level (30% D.L.L.). Subsequent loading above the 30% D.L.L. level resulted in a continuation of the nonlinear inelastic behavior in the skins. Additional load cycles to the 50% D.L.L. level yielded repeatable linear elastic response in the skins.

The initial run to 100% D.L.L. resulted in a failure at the 70% D.L.L. level. Again nonlinear inelastic behavior was exhibited by the skins once the previously high loading point was exceeded (50% D.L.L.). The failure occurred in the tension skin, a crack initiating at the corner radius, just outboard of the aft bolt hole, and propagating across the skin following a

path of minimum net section (see Figure 29).

This failure can be attributed to stress concentrations present at the corner radius which are amplified by the close proximity of a fastener. Strain levels monitored on both tension and compression skins at time of failure were similar to those predicted by analysis except in the local failure area. In addition, the load-strain and load-strain and load-deflection behavior of the specimen was highly non-linear, and large permanent deformations were present after testing at various load levels under the failure load.

Stress strain behavior of a tensile coupon cut from the same laminate as the B/Al subcomponent tensile skin is shown in Figure 30. Stress/strain data for the 6061 Al matrix is also plotted. The early onset of plasticity in the Al matrix appears to have a significant influence on the overall stress/strain response of the B/Al composite when subjected to loading. The B/Al laminate begins exhibiting inelastic behavior at approximately the same strain level that the 6061 Al becomes plastic.

FINAL WING DESIGN CRITERIA

Based on the results of this test, a review of stress-strain behavior of tensile specimens and some limited data on stress concentration in drilled holes, the following critieria was formulated for final design of the wing skins:

Nominal limit load stress \leq 360 MPa Strains at limit load \leq 2000 \mathcal{M} m/m Stress concentration factor = 1.5

It was the above design criteria which dictated the need for additional B/Al ply build up to 24 plies, in the aft attachment region, on both tension and compression wing skins to relieve stress concentrations due to attachment holes.

Final analysis using the NASTRAN finite element program was performed to confirm the stress and strain levels in the wing. The estimated total weight is 52.8 kg, 30 percent less than that of the production wing, which was designed for only 422 degrees K. Of the total weight, the skins comprise 26.3 kg, or 50 percent. The leading edge, substructure, centerline splice, and rivets and fittings weigh 5.9, 28.3, 3.2 and 4.5 kg respectively.

CONCLUSIONS

In this program, a design has been developed using metal-matrix composites to achieve high temperature capability and reduced weight. Much has been learned about the behavior of boron-aluminum and critieria for its use in aircraft structures. Additional development work would be required before it could be incorporated into an actual system. In particular, more data is

needed on fatigue and on stress concentrations in loaded holes both at low and high temperatures; basic fracture characterization should be performed; laminate tailoring should be investigated to minimize these effects as well as those due to the non-linear behavior and permanent deformations.

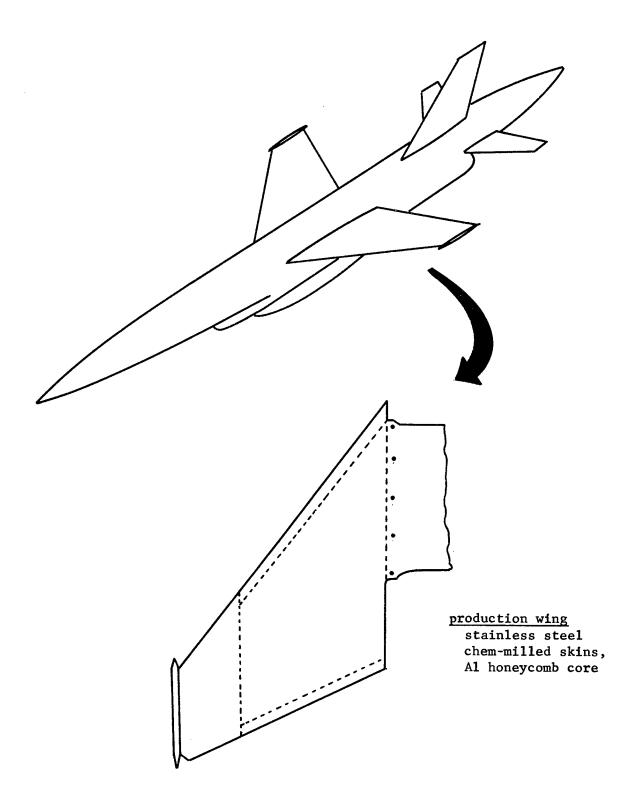


FIGURE 1 - BQM-34E RPV

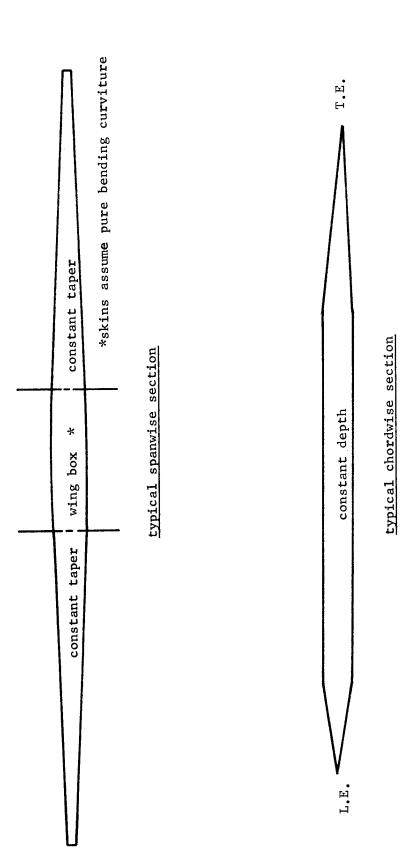
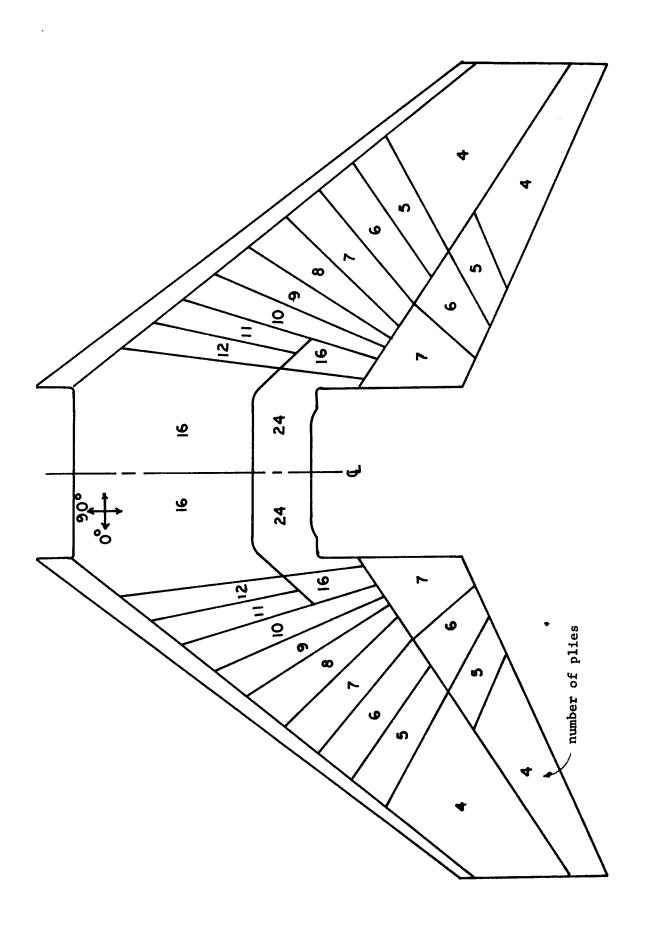
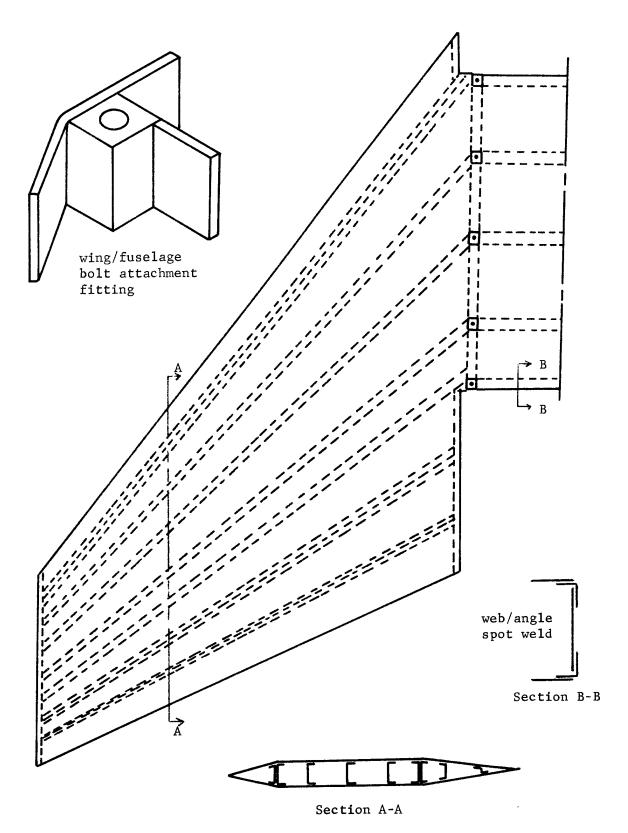


FIGURE 2 - B/AI WING SECTION GEOMETRY

FIGURE 3 - B/AI WING BASIC SKIN CONFIGURATION

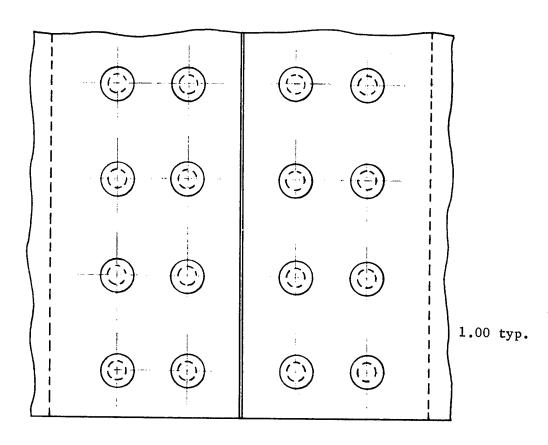
FIGURE 4 - B/AI WING TENSION SKIN LAMINATE DESIGN

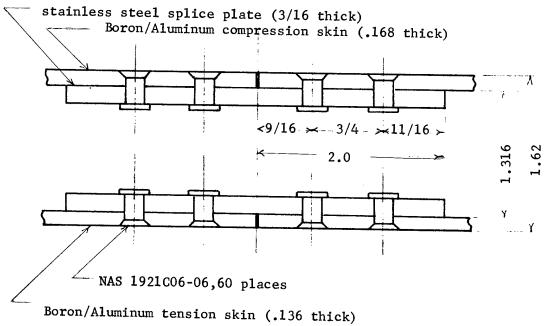


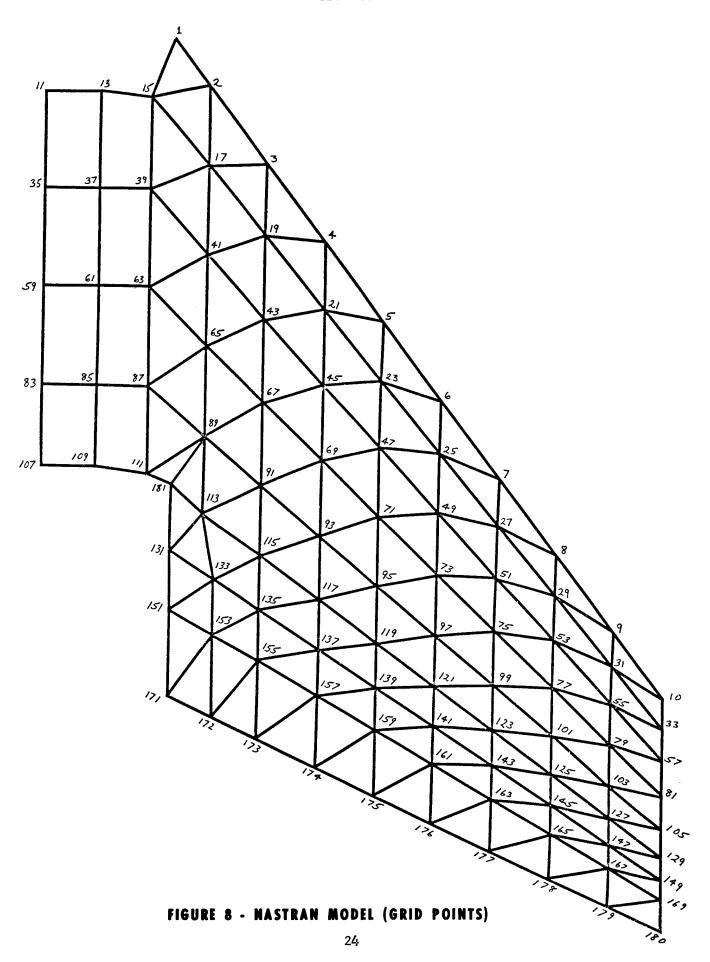


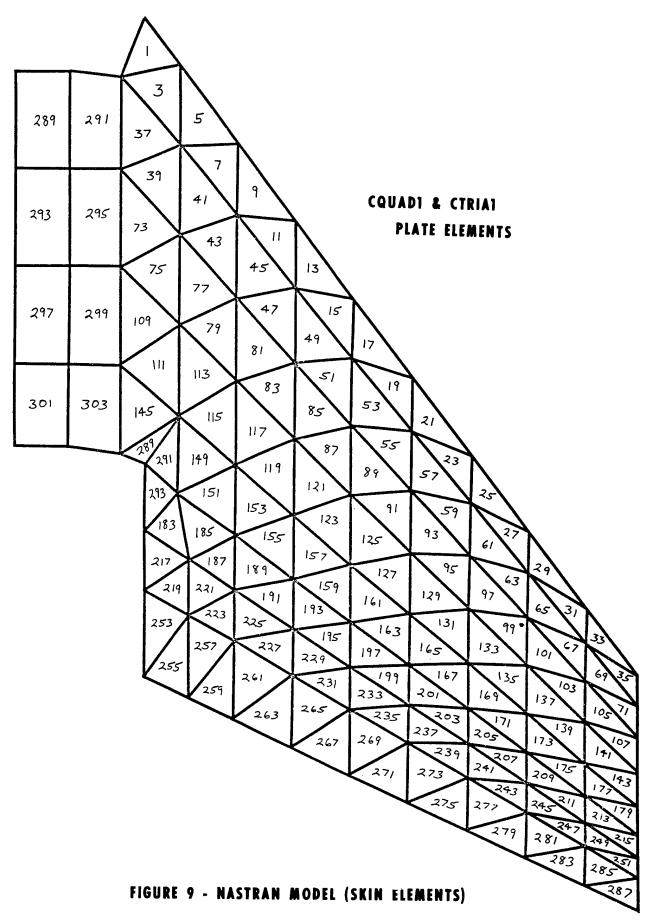
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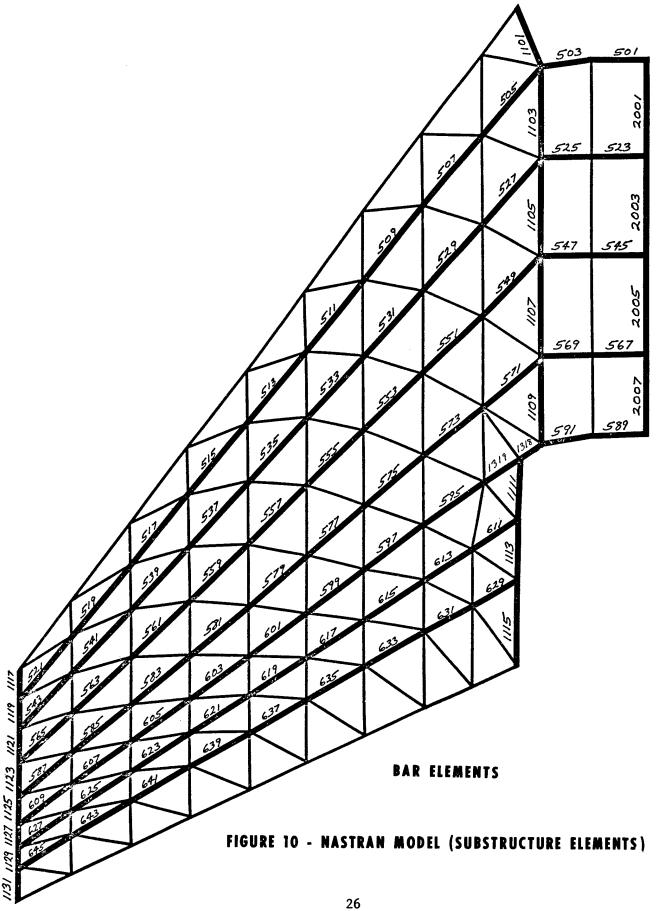
FIGURE 6 - B/AI WING SUBSTRUCTURE

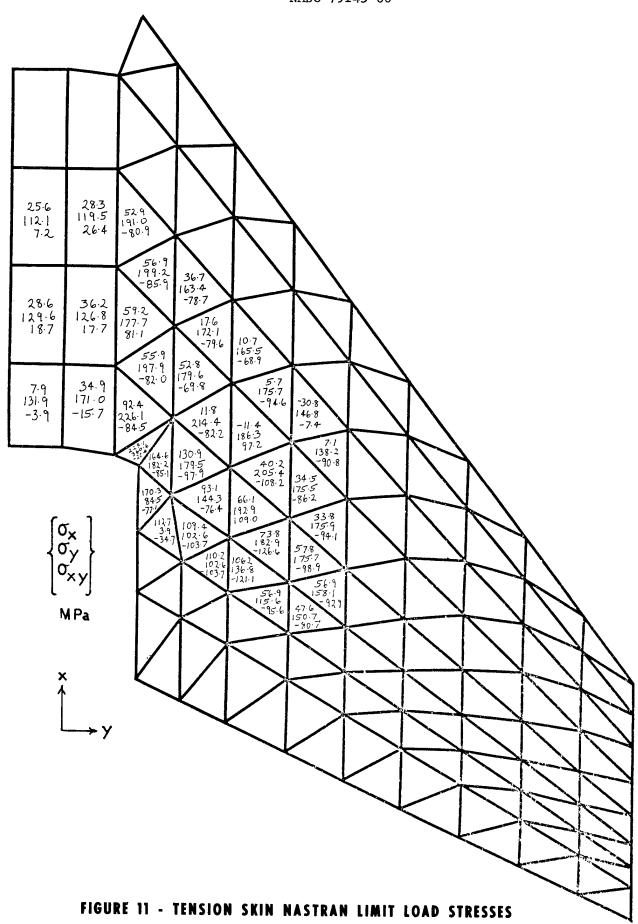


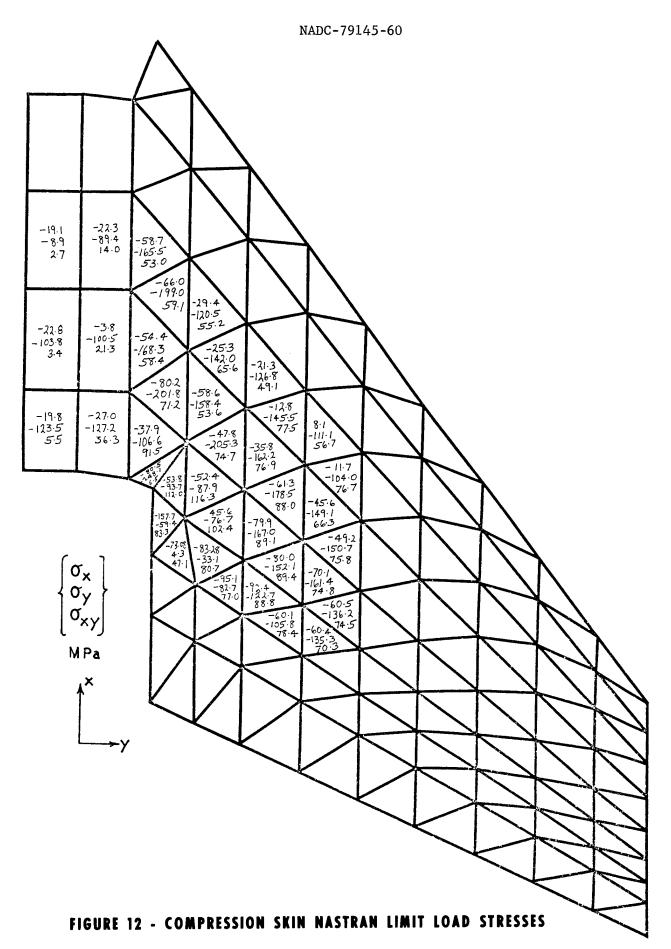


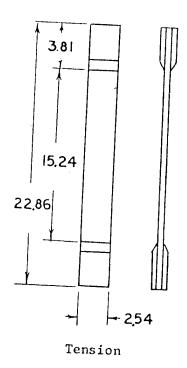


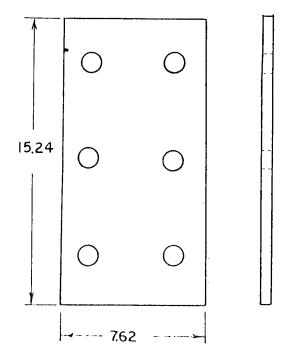












In-Plane Shear

FIGURE 13 - MATERIAL COUPON SPECIMENS

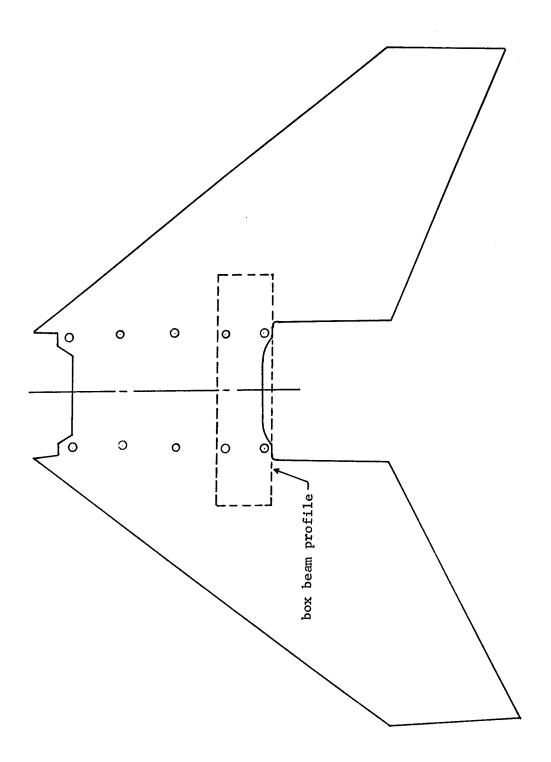
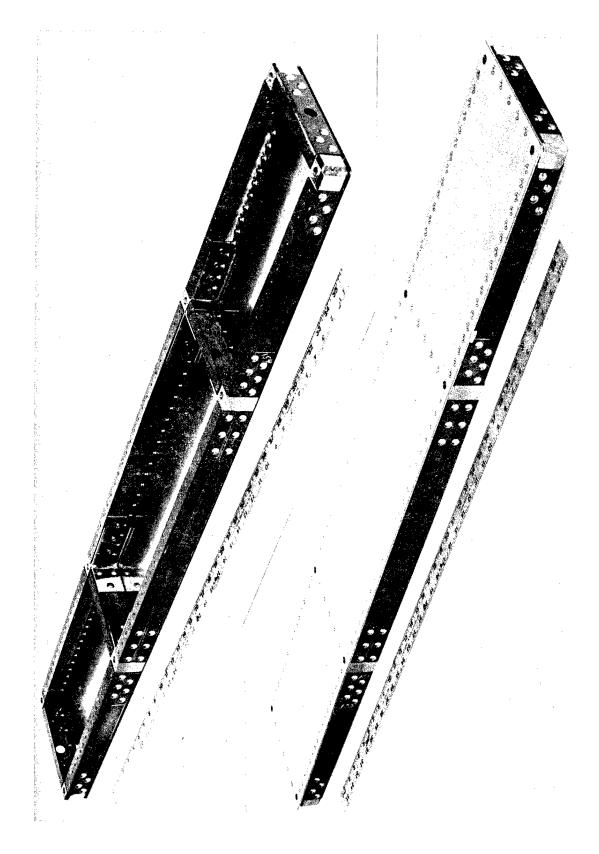
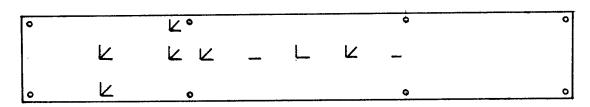
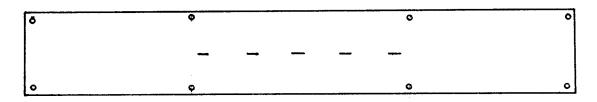


FIGURE 14 - BOX BEAM SUBCOMPONENT PROFILE

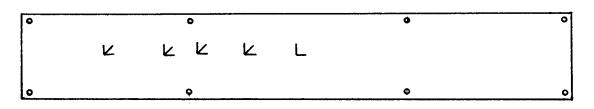




compression skin - outer surface



compression skin - inner surface



tension skin - outer surface

Key
- axial gage
- 0 / 90 gage
- rosette

Figure 16 - Box Beam Subcomponent Instrumentation

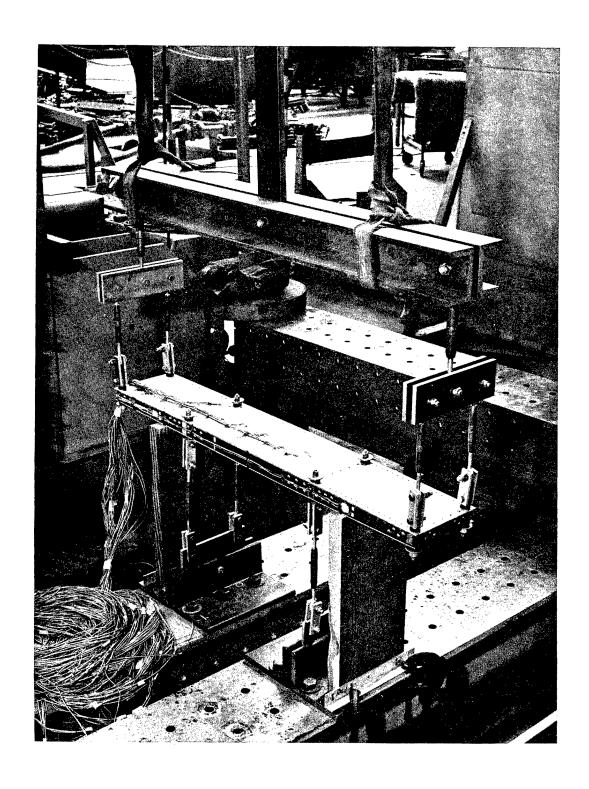
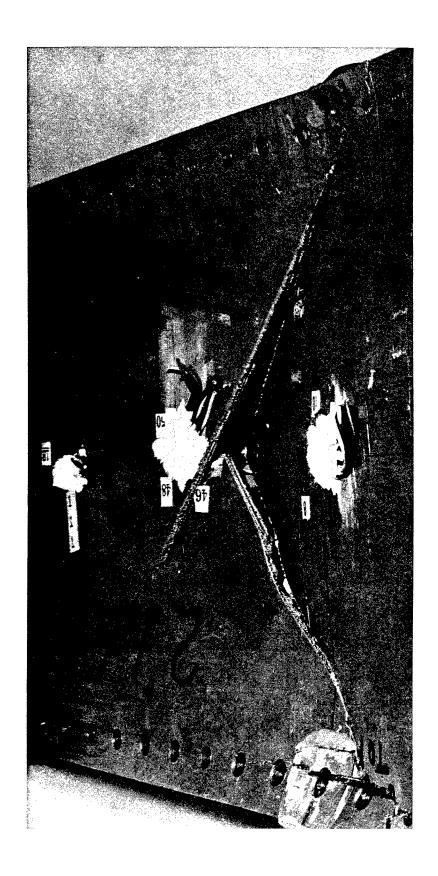
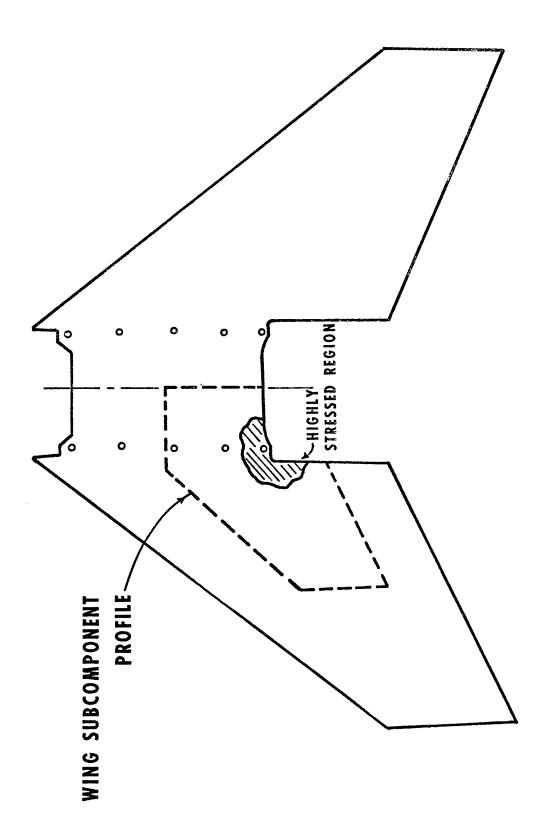
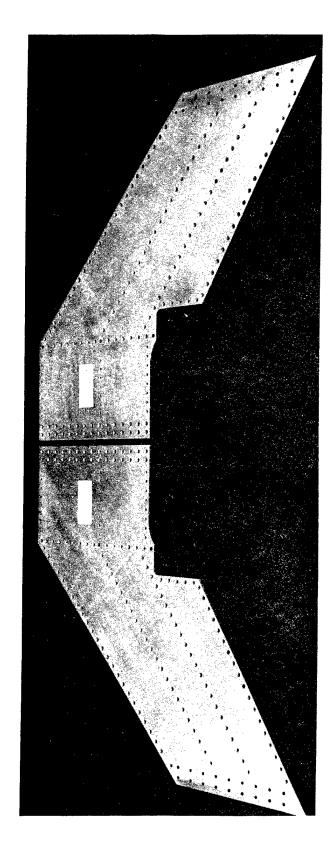


FIGURE 17 - BOX BEAM SUBCOMPONENT TEST SETUP



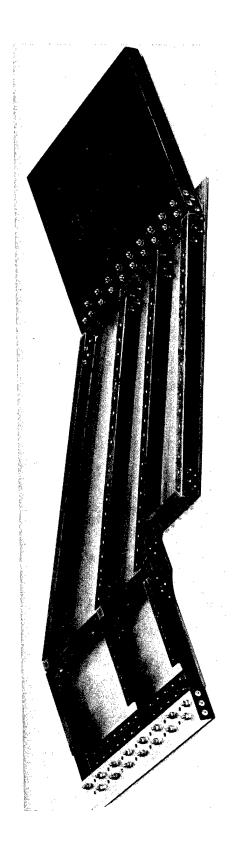




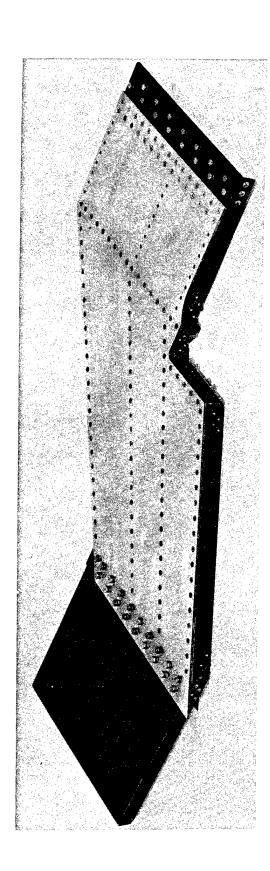
COMPRESSION SKIN - 16 PLIES

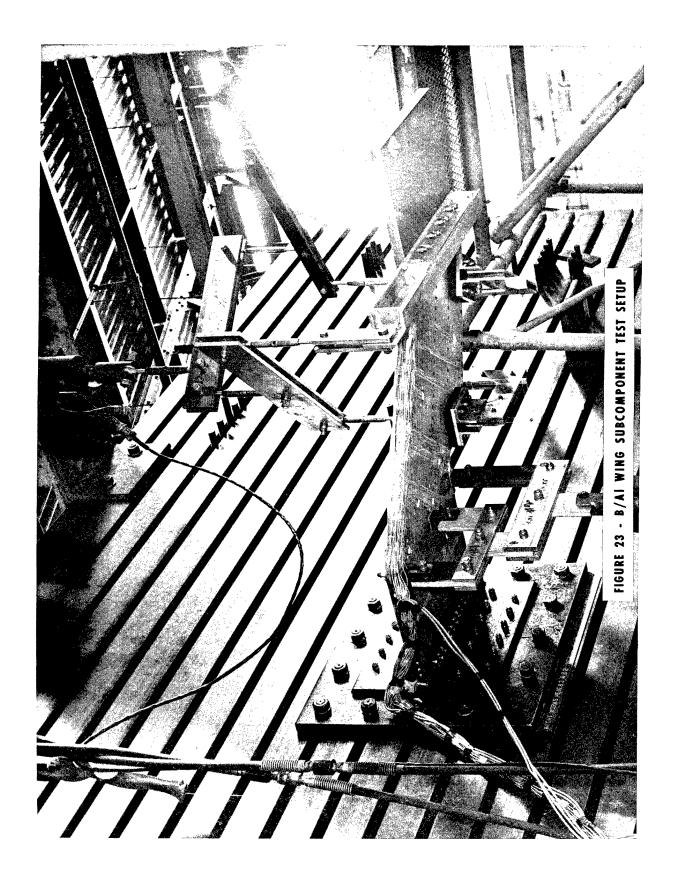
TENSION SKIN - 13 PLIES

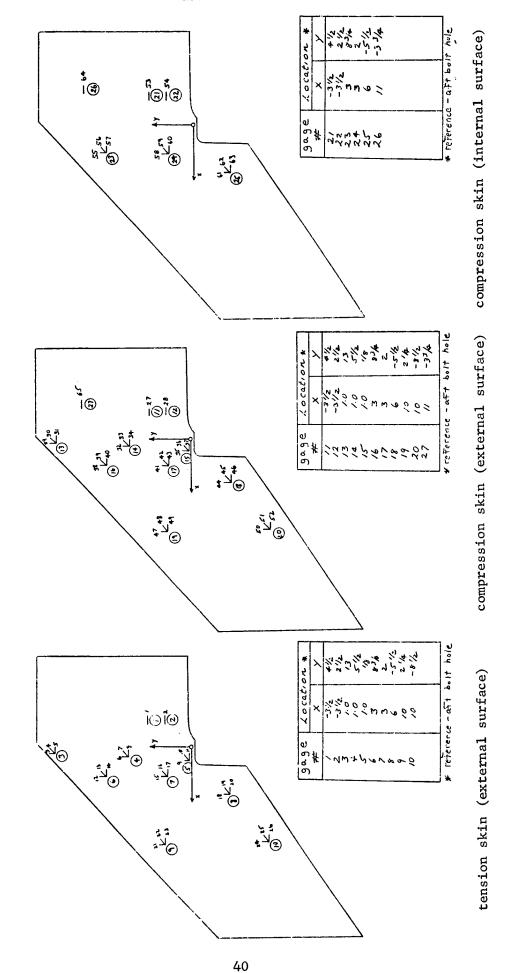
FIGURE 20 - B/A! WING SUBCOMPONENT SKINS











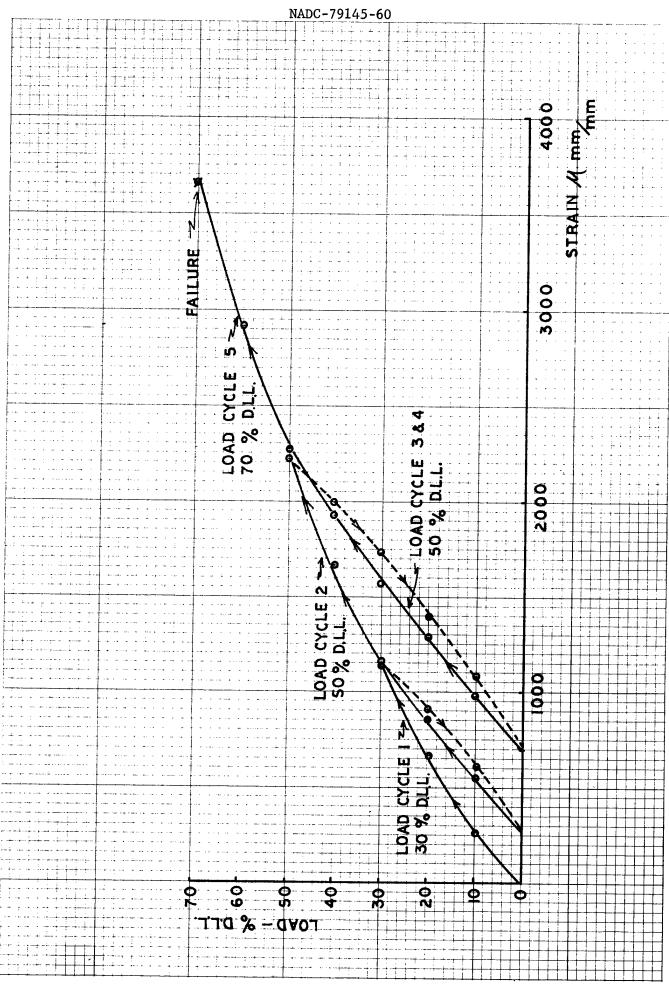
- rosette

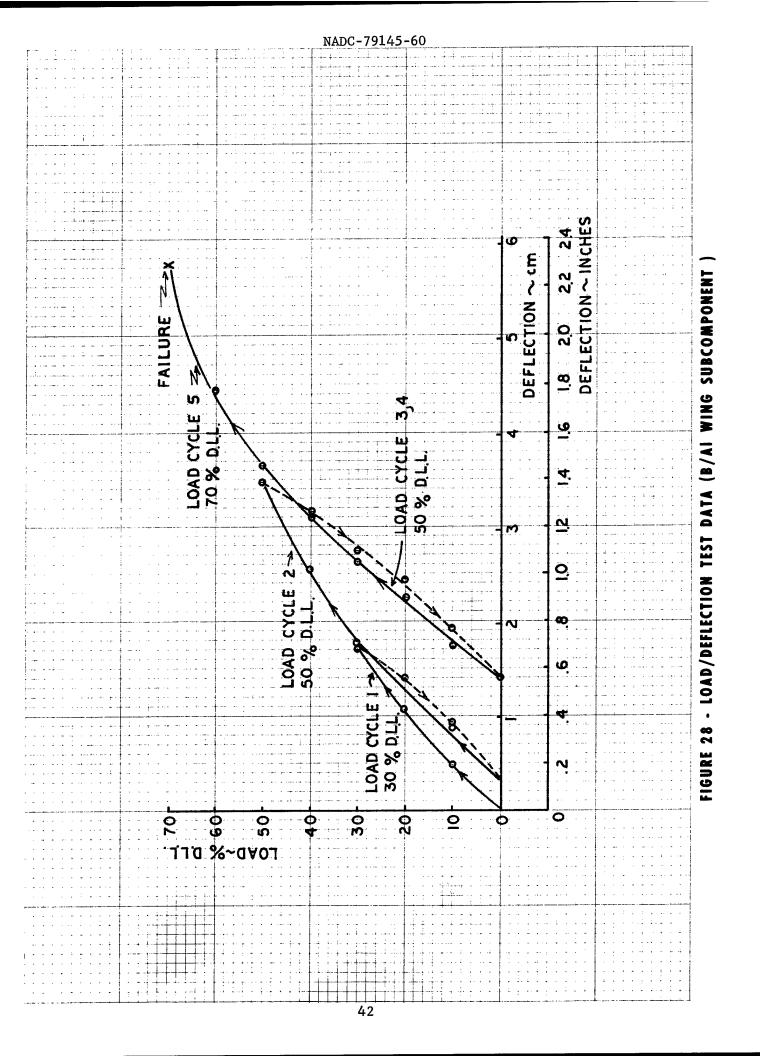
لا

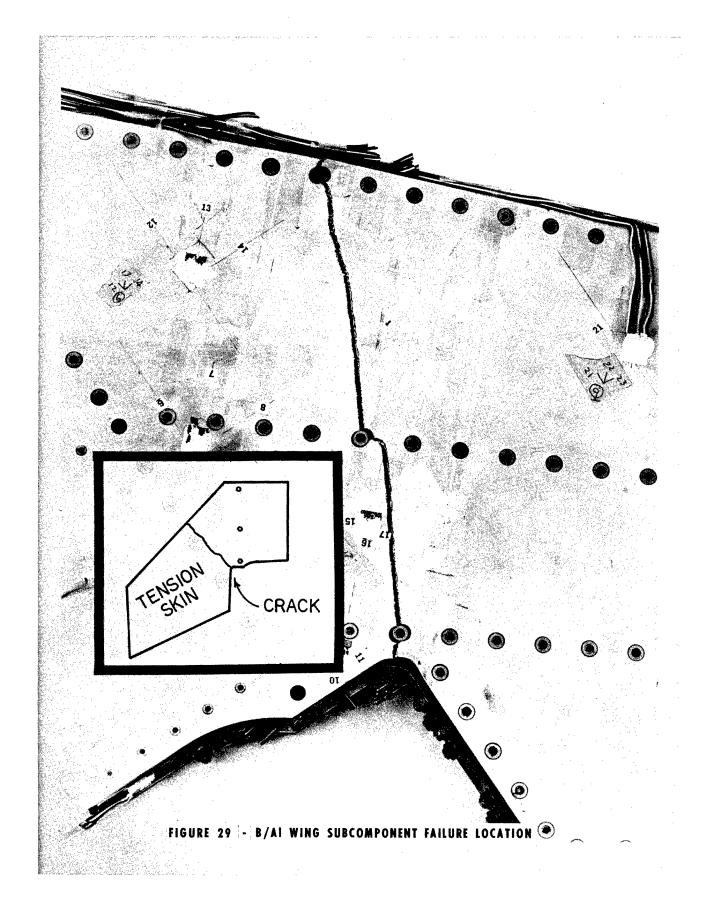
- axial gage

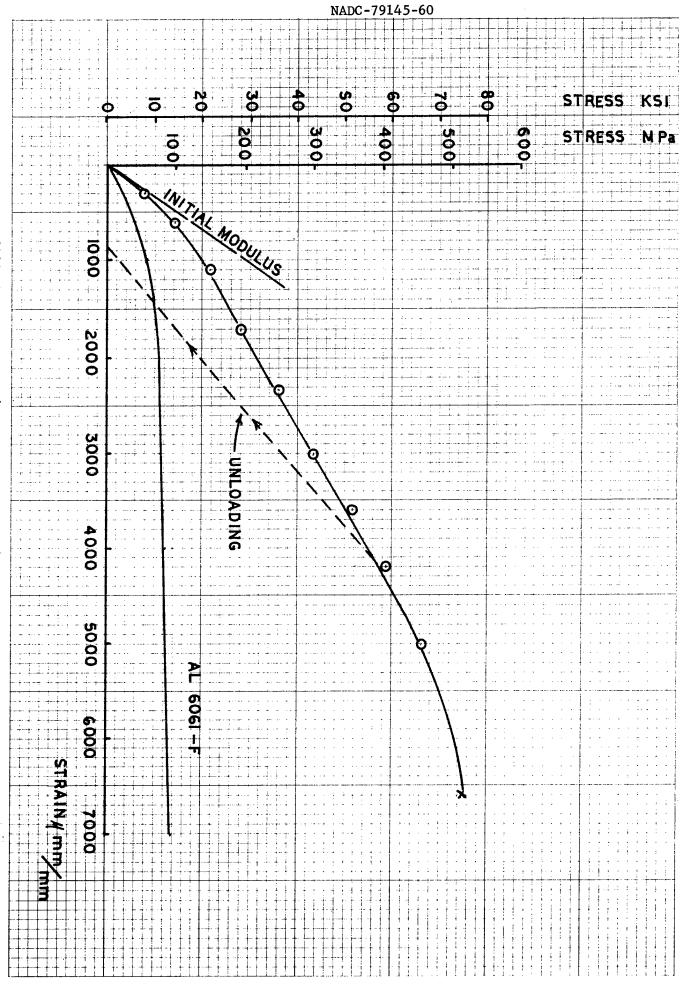
KEY

Figures 24,25 & 26 - B/Al Wing Subcomponent Instrumentation









NASTRAN Element Number	Nxult KN/m	Nxyult KN/m	Nxcr KN/m	Nxycr KN/m	<u>Nxult</u> Nxcr	<u>Nxyult</u> Nxycr	Margin of Safety
290	-1985	15.6	-9619	13518	.205	.001	3.84
292	-750	896	-7996	13518	.094	.066	6.81
294	-1261	666	-1737	17764	.746	.037	.37
150	-469	620	-3055	5263	.154	.118	3.60
116	-822	299	-800	1072	1.028	.279	09
154	-501	267	-698	1091	.718	. 245	.26
156	-409	547	-4619	6827	.089	.080	6.36
158	-377	175	-425	641	.887	.273	.04
190	-327	237	-406	808	.805	.293	.11
194	-271	140	-265	403	1.023	. 347	11
162	-284	110	-294	428	.966	.257	03
198	-223	88	-251	344	.888	. 256	.05
304	-1020	291	-4588	6094	. 223	.048	3.31
166	-216	66	-211	289	1.024	.228	07

Table 1 - B/Al Wing Compression Skin Critical Buckling Loads

# Specimens	Ult. Tensile Stress (MPa)
5	492.7
5	189.3
6	327.3
	5 5

Table 2 - Results Tensile Coupon Tests

Laminate Type	# Specimens	Ult. Shear Stress (MPa)
0/ <u>+</u> 45	5	257.4
0	5	131.4
<u>+</u> 45	5	309.6

Table 3 - Results Rail Shear Coupon Tests

Laminate Type	E ₁ (GPa)	E ₂ (GPa)	G (GPa)	12	21
0/ <u>+</u> 45	158.2	131.3	50.5	.331	.307
<u>+</u> 45	137.2	137.2	54.9	.364	.364

Table 4 - Experimental Material Property Constants

p	
Fibers in Test Direction (%)	23.1 23.1 30.1 30.1 18.8 18.8 31.3
Failure Strain m/m	.0080 .0075 .0066 .0066 .0088 .0077
Ult. Stress MPa	541.9 530.9 532.3 612.3 484.7 450.2 528.8 504.0
Ult. Load KN	23.8 23.5 23.9 27.9 24.5 24.5 27.2
Prop. Limit MPa	88.3 86.2 97.9 95.2 53.8 51.0 75.8
E ₁ GPa	175.8 173.1 120.0 157.2 151.0 134.5 163.4
X-Sect Area (cm ²)	.445 .448 .447 .462 .554 .551 .557
Specimen Number	6240P-A1 6240P-A2 6240P-A3 6240P-A4 6241P-A1 6241P-A2 6241P-A3 6241P-A4
Test Dir.	06 0 06 06 06
Wing Skin	777777777777777777777777777777777777777

(Tension) ----(0,90,+45,-45,0,-45,90,+45,0,+45,-45,90,0) 13 ply Wing Skin -1

Wing Skin -2 (Compression)----(0,90,+45,-45,0,-45,+45,0,0,+45,-45,90,-45,+45,90,0) 16 ply

APPENDIX A

NASTRAN BULK DATA

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NASTRAN

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APP DISPLACEMENT
-SOL 1,0
-IMC 15
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B/AL WING STATIC ANALYSIS, EXP. PROP. Skin changes of 10-31-73 + New G11 of elem

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	9 . 10	+PAR1114	+PAR1115	-0000+P2R1116	+PAR1117	+PA31118 +0000+P2R1118	+PAR1119	+PAR1120	+PAR1121	.0000+P2R1122	+PAR1123 +0000+P2R1123	+PAR1124 •0000+P2R1124	+PAR1125 +0000+P2E1125	+PAR1126	+PAR1127	+P4R1128	+PA41129
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B/AL WING STATIC ANALYSIS, EXP. PROP. SKIN CHANGES OF 10-31-78 + NEW GII OF ELEM 289+290(11-2-78)

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. 6 10	+PAR1315	+PAR1316	+PAR2001			120+	+	*0 +T149	•0. +T289	0.0 +1011	0.0 +1104	0.0		0.1			0.0 +1109	0.0 +1110	0.0+T113 -	0.0	0 +T20	0-1-0-	0.4120	07-
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NO ERRORS FOUND - EXECUTE NASTRAN PROGRAM

*	SYSTEM	INFORMATION	MESSAGE	3107.	*** SYSTEM INFORMATION MESSAGE 3107. EMGOLD IS PROCESSING ELEMENTS OF TYPE = 34, BEGINNING WITH ELEMENT ID =	:	501
*	SYSTEY	INFURMATION	MESSAGE	3113,	*** SYSTEM INFORMATION MESSAGE 3113, ENGPRO PROCESSING SINGLE PRECISION ELEMENTS OF TYPE 10	10 STARTING WITH ID	-
*	SYSTE*	"INFORMATION	T MESSAGE	3113,	*** SYSTEY INFORMATION MESSAGE 3113, EMSPRO PROCESSING SINGLE PRECISION ELEMENTS OF TYPE 10	19 STARTING WITH ID 289	. 583
*	*** SYSTE4	INFORMATION	MESSAGE	3107.	*** SYSTEM INFORMATION MESSAGE 3107. EMBOLD IS PROCESSING ELEMENTS OF TYPE = 19, BEGINNING WITH ELEMENT ID =		289
*	SYSTEM	INFORMATION	NESSAGE	3113,	*** SYSTEM INFORMATION MESSAGE 3113, E4GPRO PROCESSING SINGLE PRECISION ELEMENTS OF TYPE	6 STARTING WITH ID	-

= 5, BEGINNÍNG WITH ELEMENT ID = 1, EST. TIME = 1.4
1,EST. TIME = .1

****SYSTEM"INFORMATION MESSAGE 3107." EMSOLD'IS PROCESSING ELEMENTS OF TYPE

METHOD 1 NT.NBR PASSES =

METHOD 3 T .NBR PASSES =

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